US010501200B2

(12) United States Patent
Ullyott et al.

(10) Patent No.: US 10,501,200 B2
(45) Date of Patent: *Dec. 10, 2019

(54) **ENGINE ASSEMBLY FOR AN AUXILIARY POWER UNIT**

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(*) Notice: Subject to any disclaimer, the term of this patent is extended or adjusted under 35 U.S.C. 154(b) by 0 days.

This patent is subject to a terminal disclaimer.

(21) Appl. No.: **16/004,991**

(22) Filed: **Jun. 11, 2018**

(65) **Prior Publication Data**

US 2018/0346142 A1 Dec. 6, 2018

Related U.S. Application Data

(63) Continuation of application No. 15/616,187, filed on Jun. 7, 2017, now Pat. No. 9,994,332, which is a (Continued)

(51) **Int. Cl.**
B64D 41/00 (2006.01)
F01C 1/22 (2006.01)
(Continued)

(52) **U.S. Cl.**
CPC ***B64D 41/00*** (2013.01); ***F01C 1/22*** (2013.01); ***F01C 11/008*** (2013.01); ***F02C 3/10*** (2013.01); ***H02K 7/1815*** (2013.01); ***H02K 7/1823*** (2013.01); *B64D 2013/0611* (2013.01); *Y02T 50/44* (2013.01); *Y02T 50/53* (2013.01)

(58) **Field of Classification Search**
USPC 290/52; 60/770–779, 612, 624; 123/216
See application file for complete search history.

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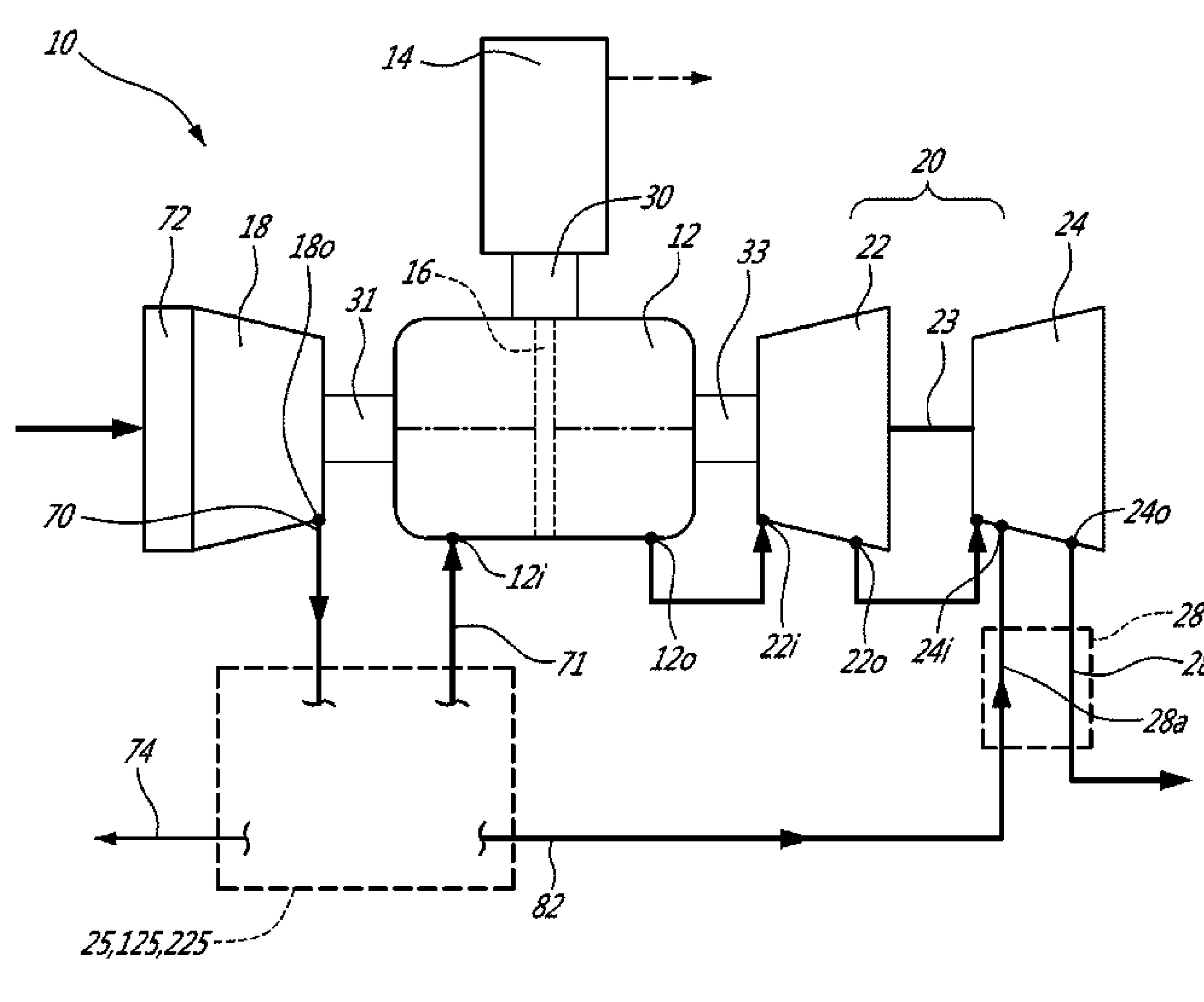
(57) **ABSTRACT**

An auxiliary power unit having internal combustion engine (s) in driving engagement with an engine shaft, a generator having a generator shaft drivingly engaged to the engine shaft, a compressor having an outlet in communication with the internal combustion engine inlet, and a turbine having an inlet in communication with the internal combustion engine outlet. The turbine may be a first stage turbine, and the assembly may include a second stage turbine having an inlet in communication with the first stage turbine outlet. A method of providing electrical power to an aircraft is also discussed.

18 Claims, 11 Drawing Sheets



Related U.S. Application Data

continuation of application No. 14/750,207, filed on Jun. 25, 2015, now Pat. No. 9,771,165.

(51) **Int. Cl.**

F02C 3/10	(2006.01)
H02K 7/18	(2006.01)
F01C 11/00	(2006.01)
B64D 13/06	(2006.01)

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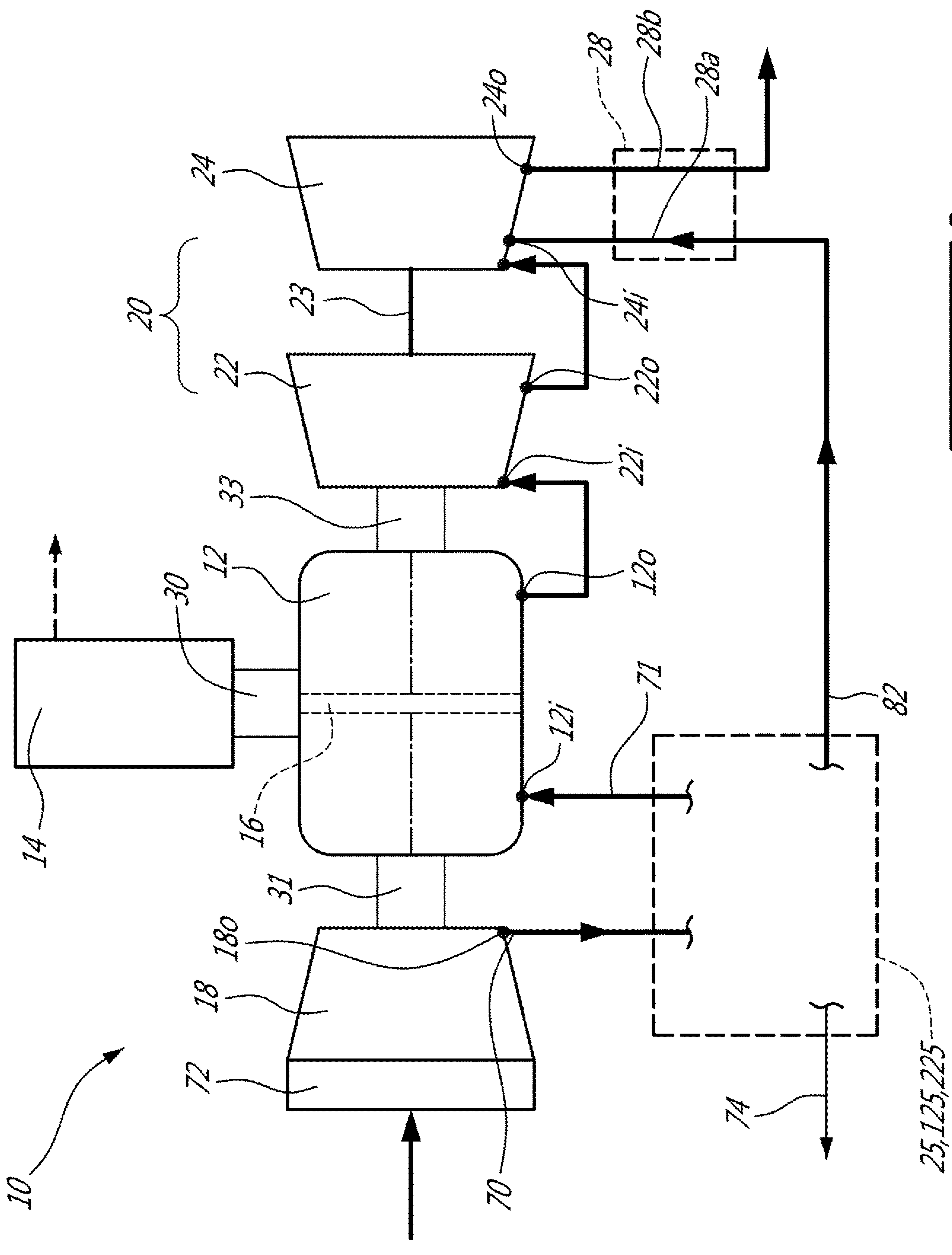
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FIG. 1

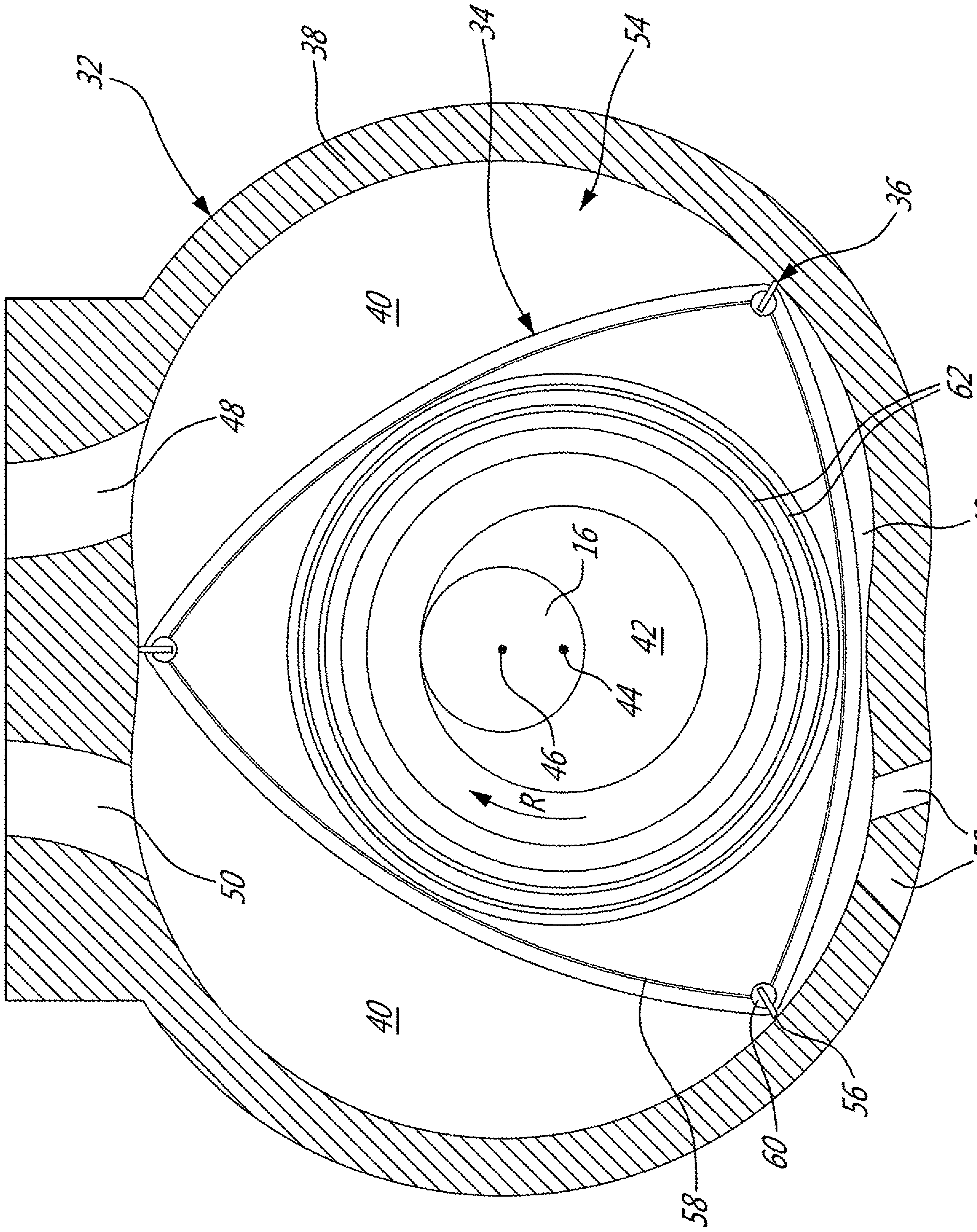

FIG-2

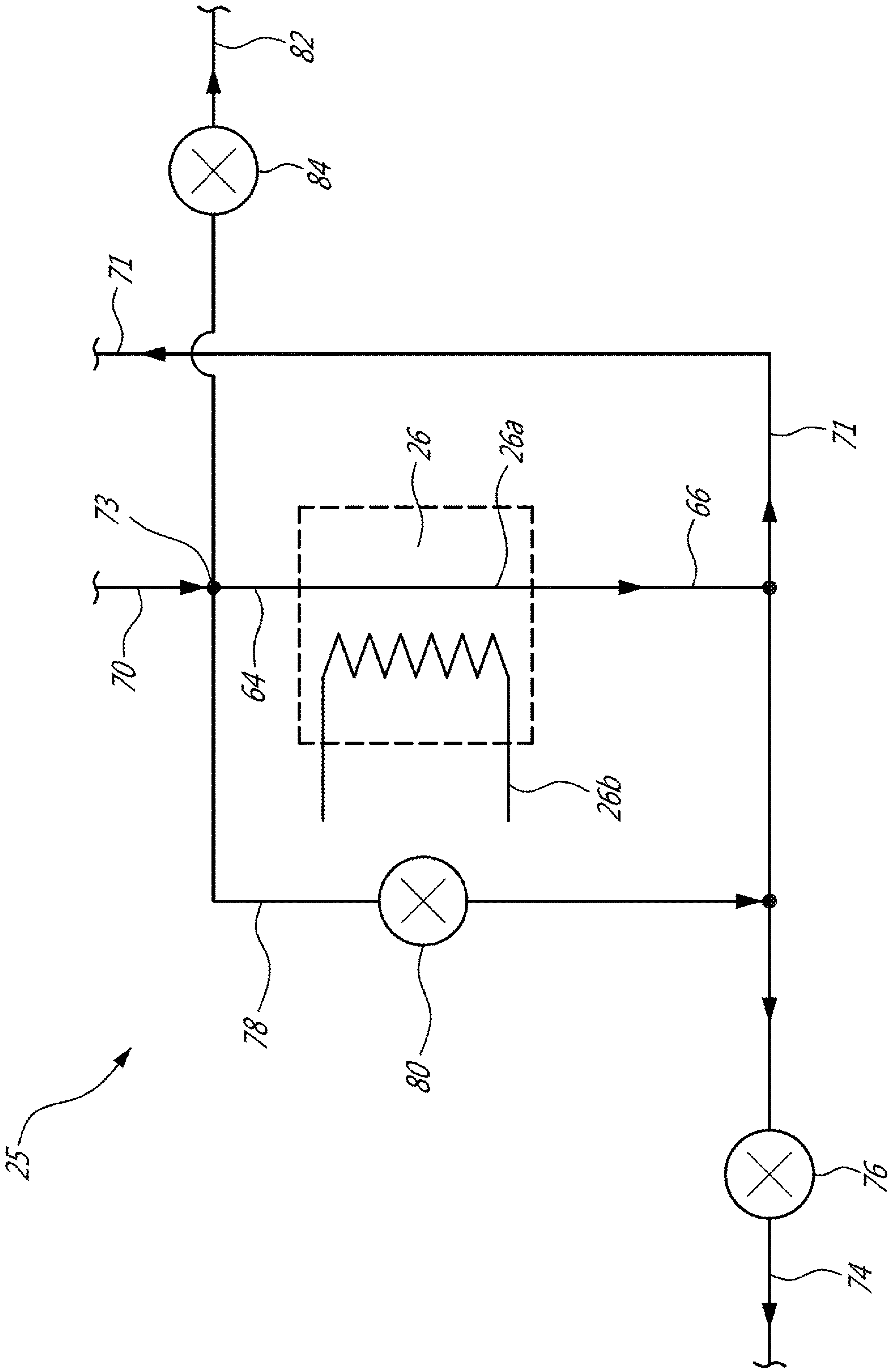

FIG-3

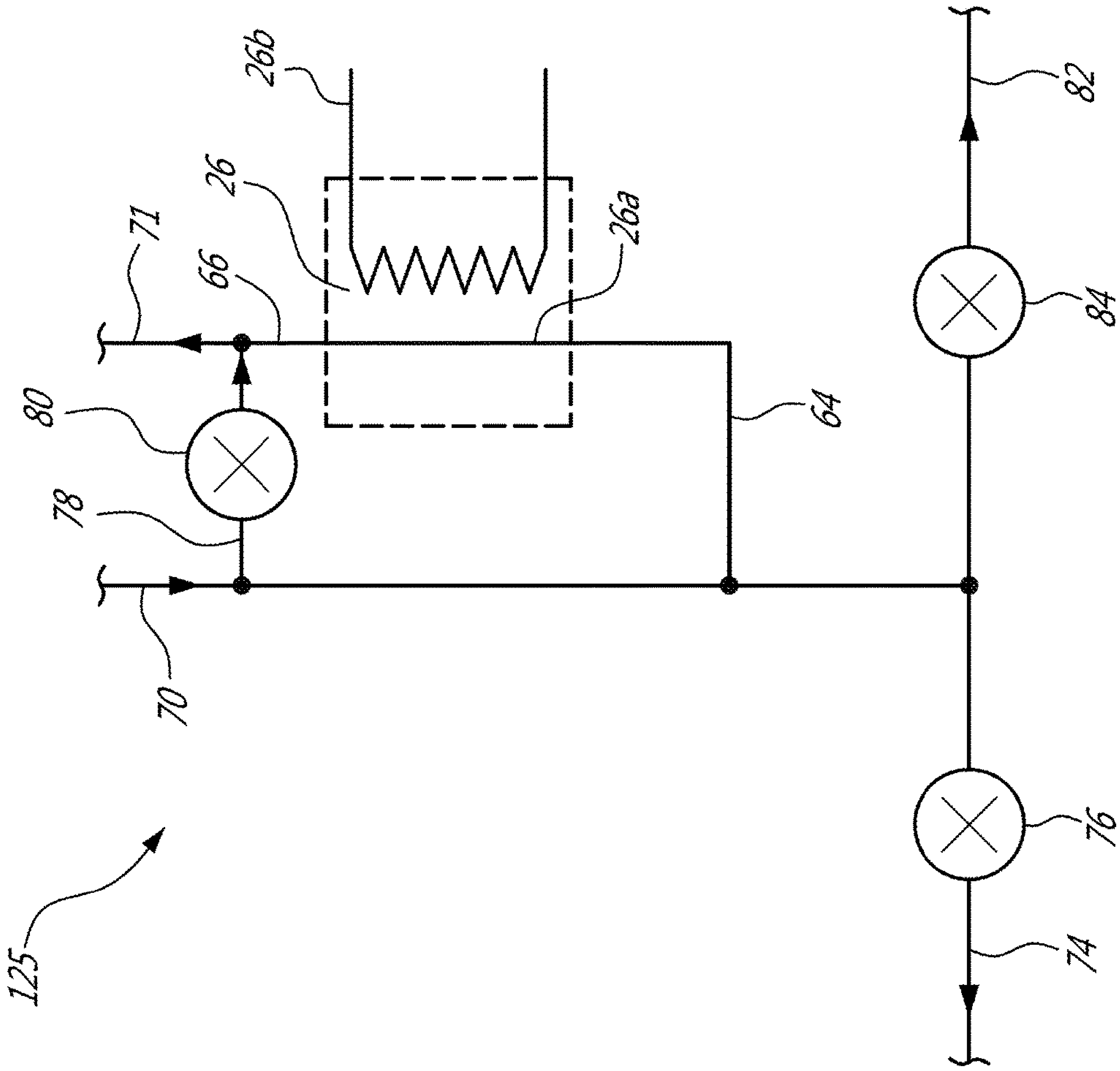

FIG. 4

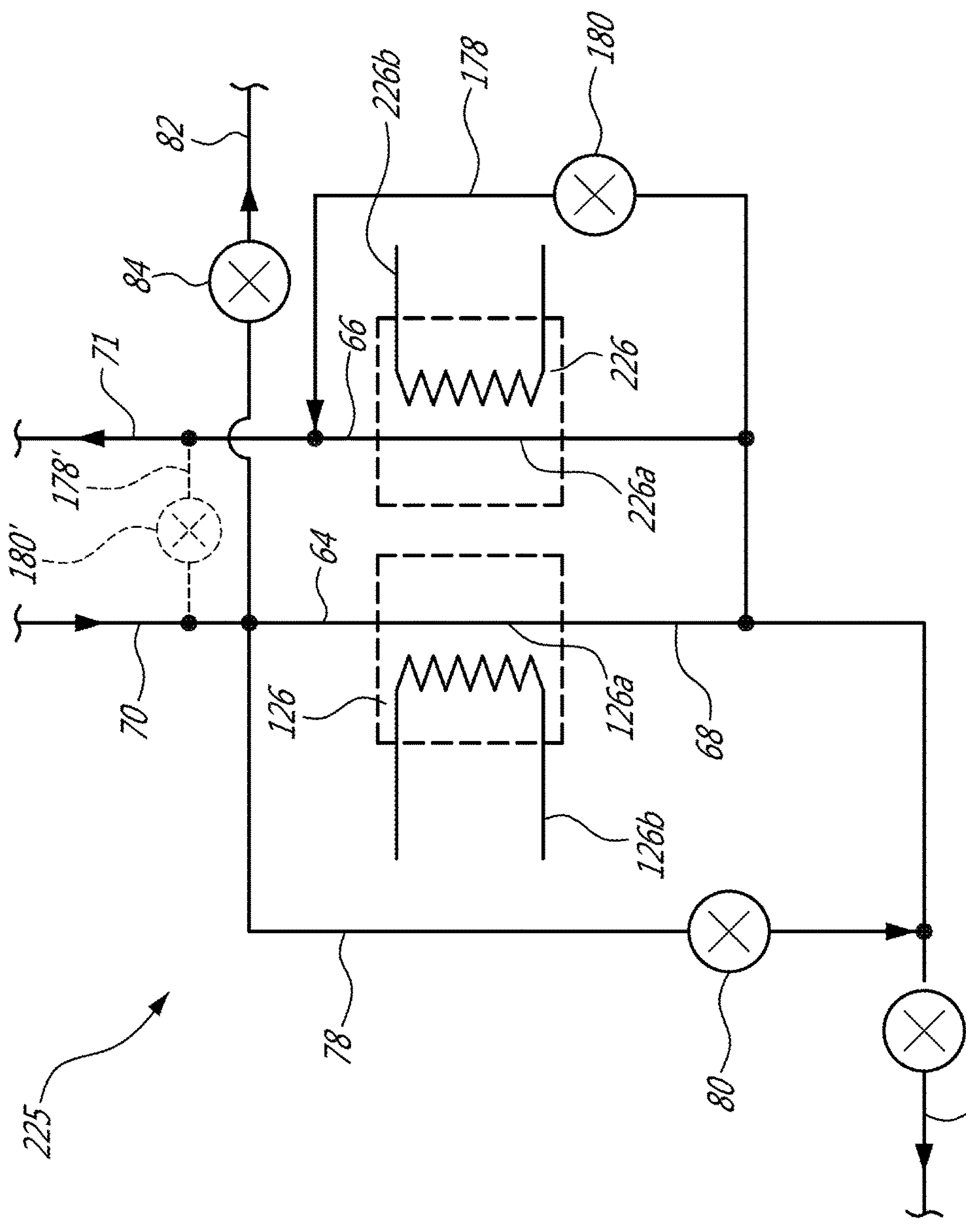
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FIG - 5

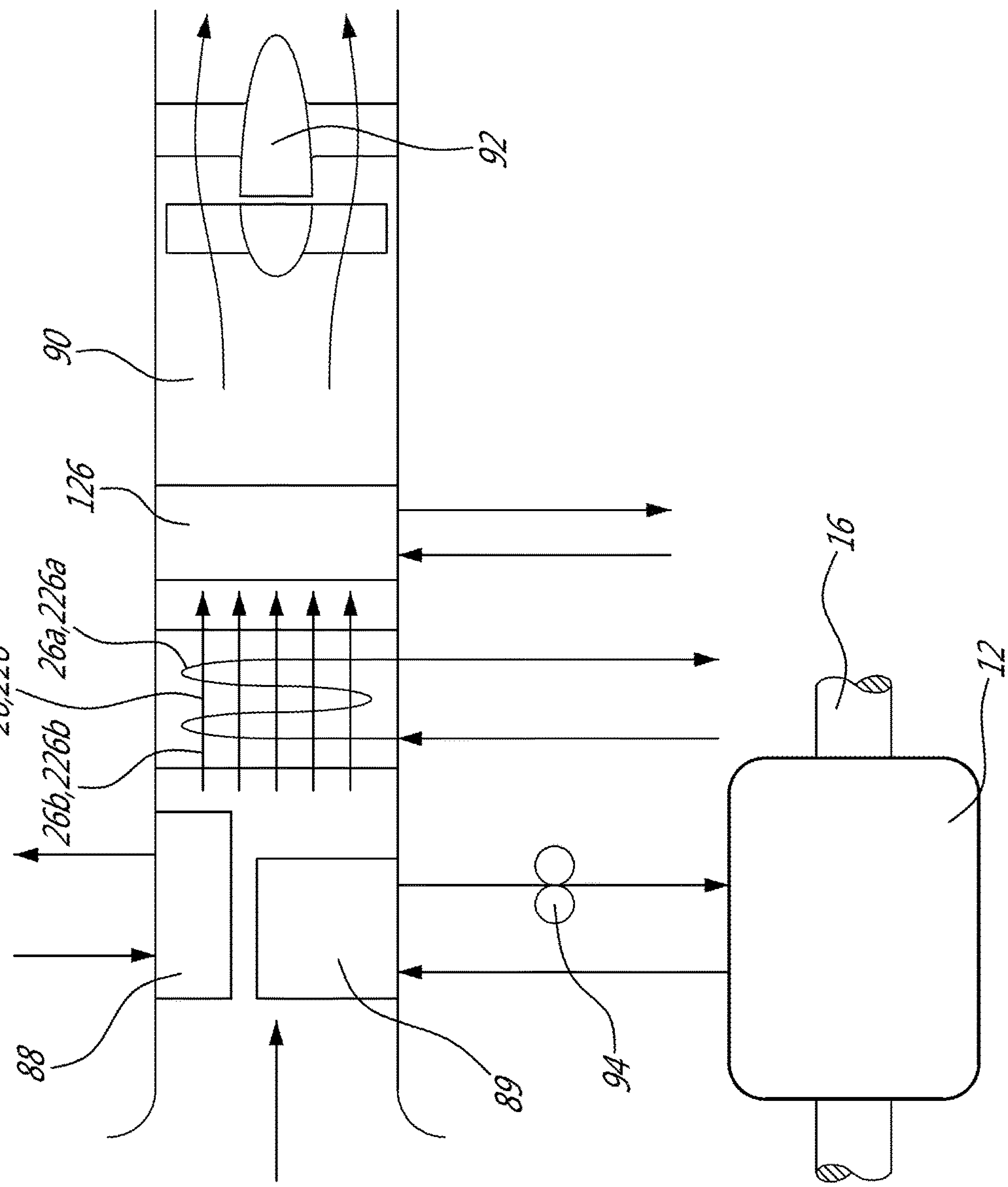

FIG-6

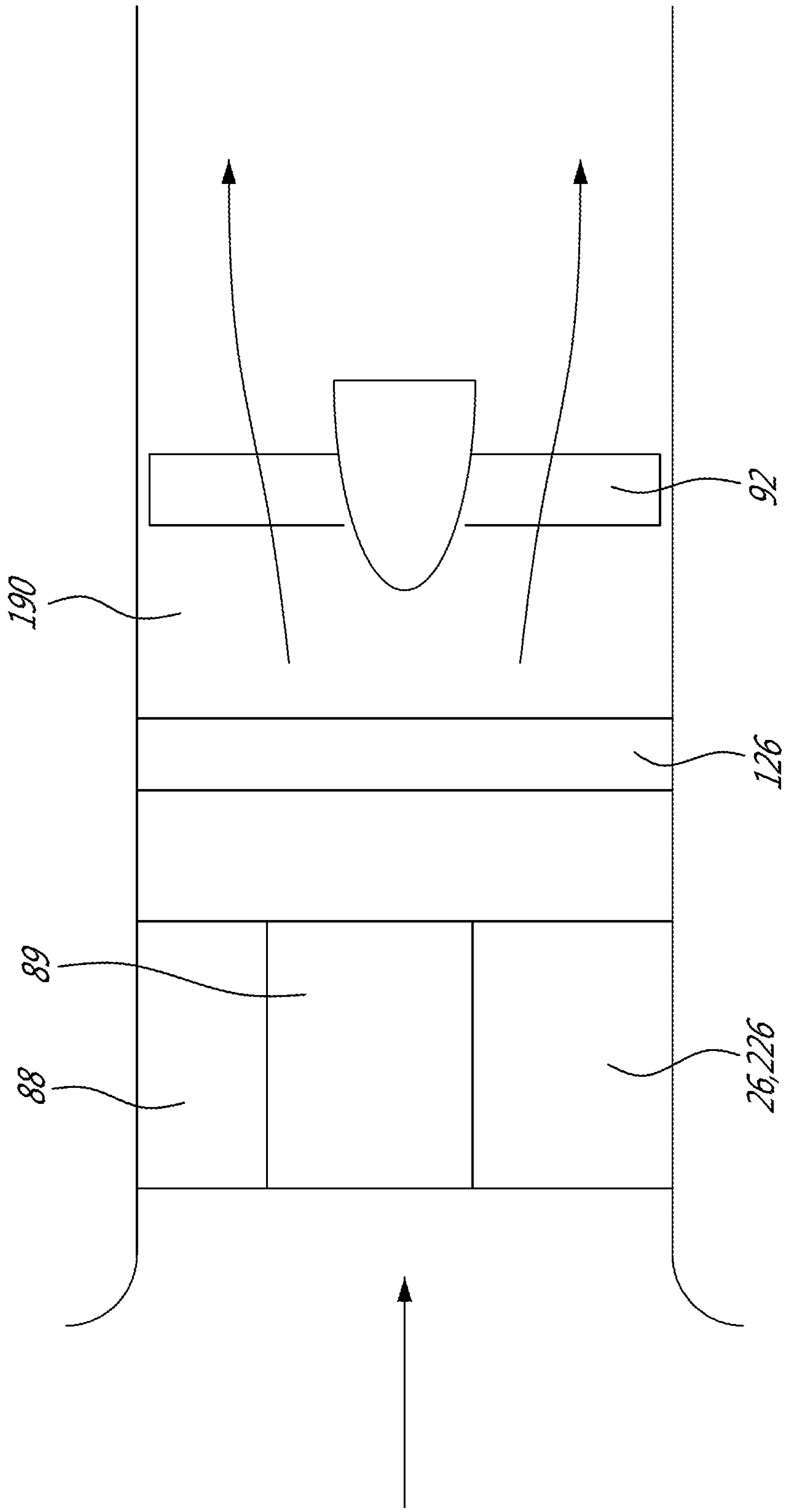

FIG. 7

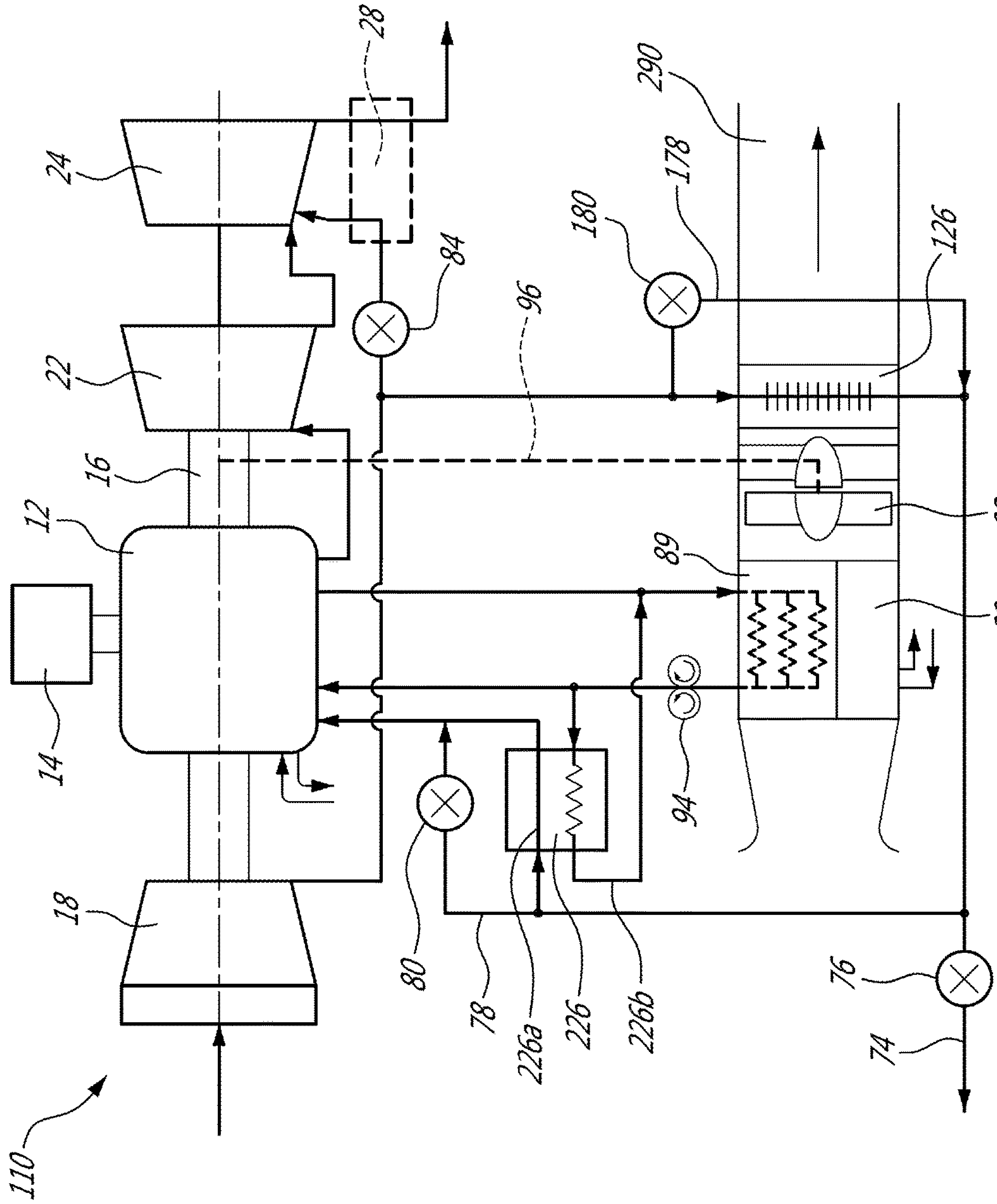
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FIG-8

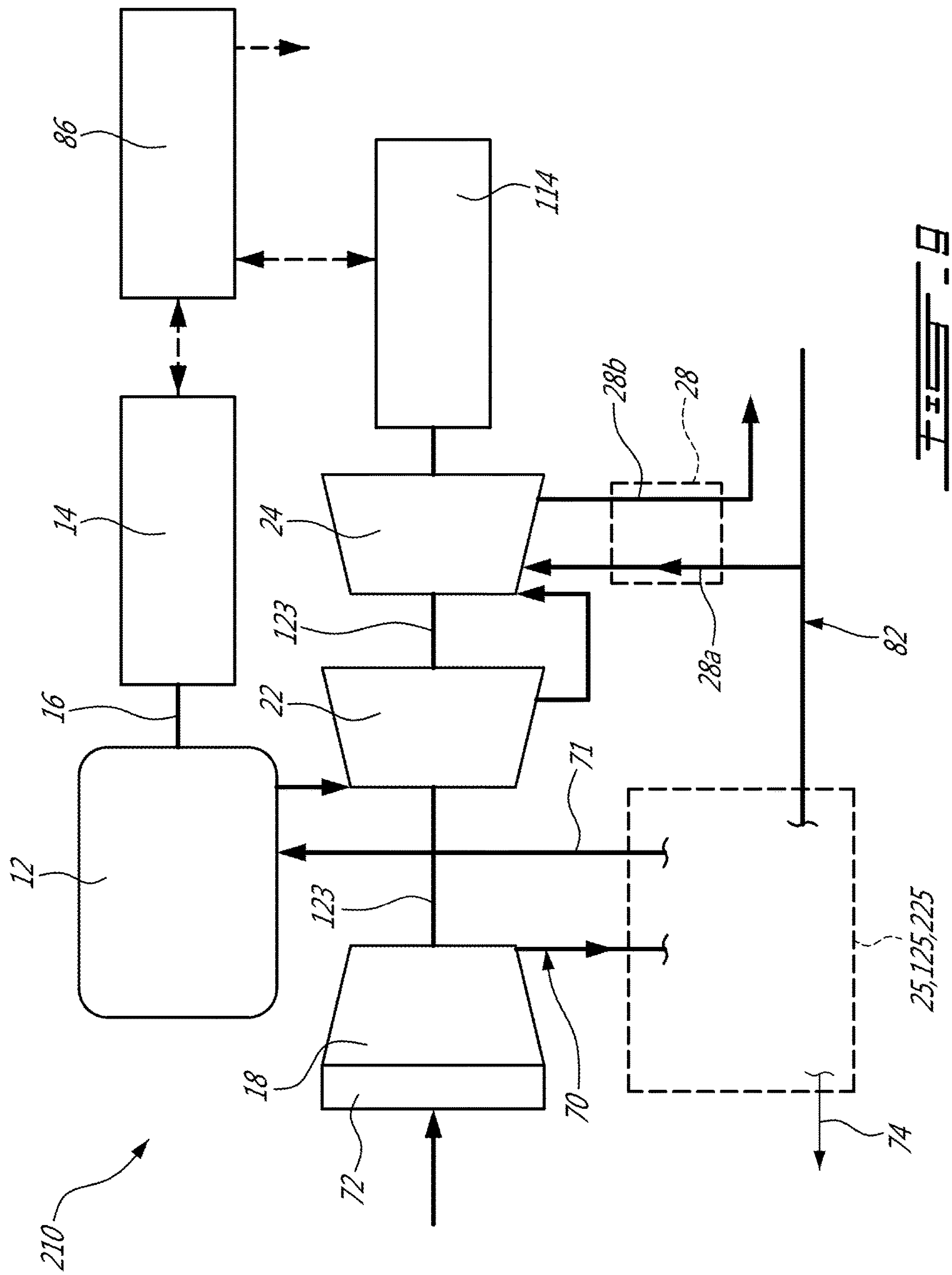
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FIG. 9

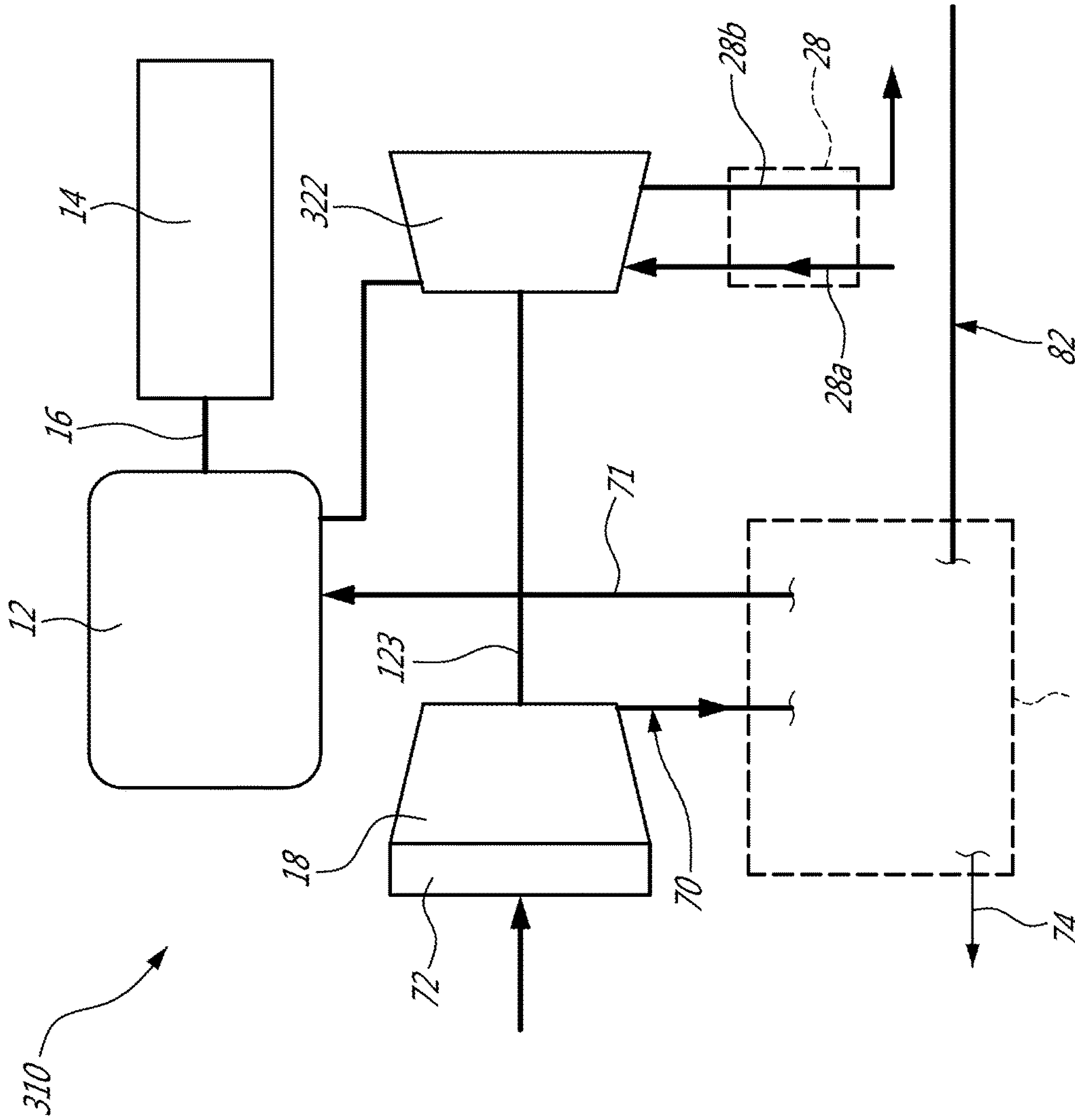

FIG-10

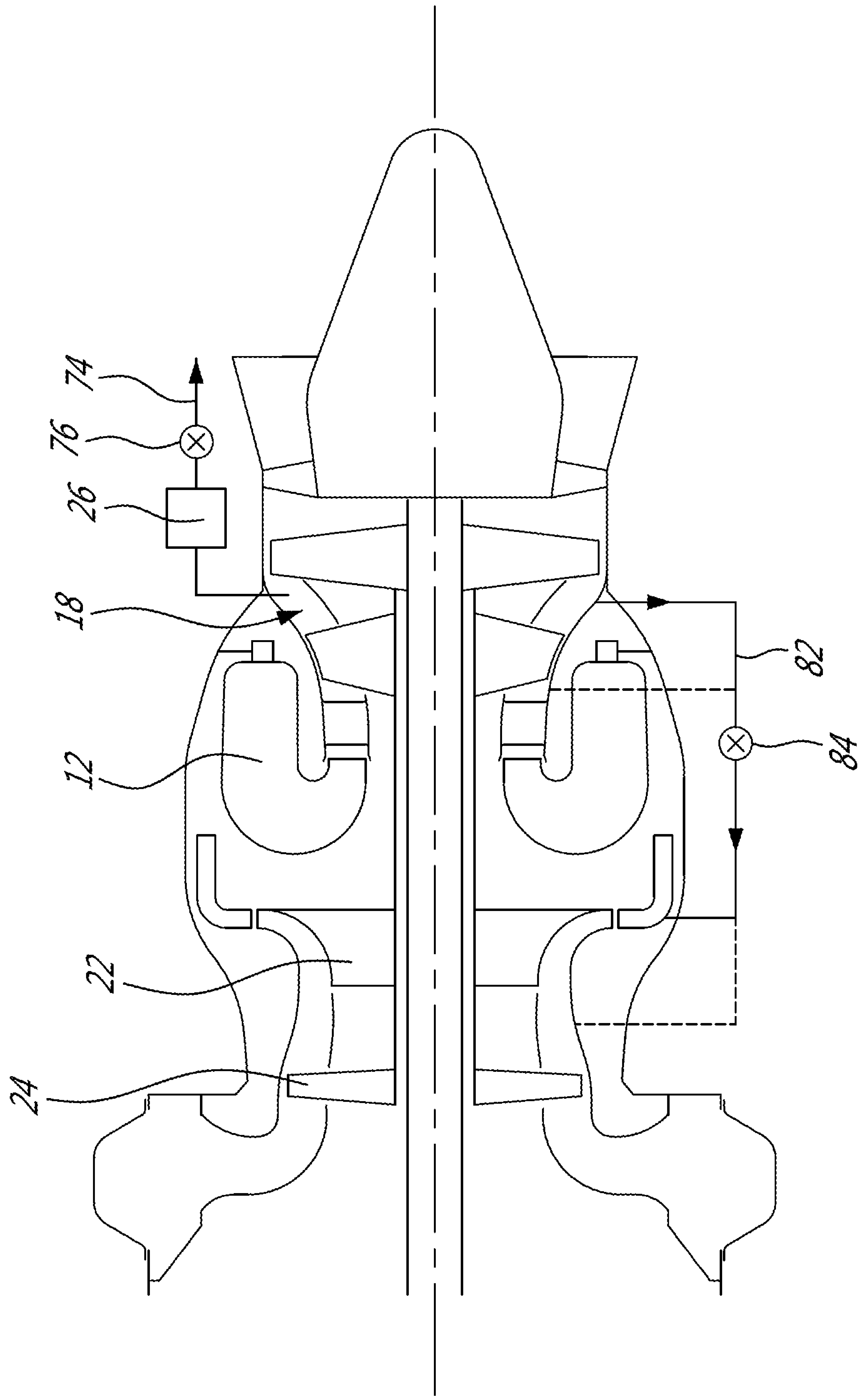

FIG-11

ENGINE ASSEMBLY FOR AN AUXILIARY POWER UNIT

CROSS-REFERENCE TO RELATED APPLICATIONS

This application is a continuation of U.S. application Ser. No. 15/616,187 filed Jun. 7, 2017, which is a continuation of U.S. application Ser. No. 14/750,207 filed Jun. 25, 2015, the entire contents of both of which are incorporated by reference herein

TECHNICAL FIELD

The application related generally to engine assemblies and, more particularly, to such engine assemblies used as auxiliary power units in aircraft.

BACKGROUND OF THE ART

Aircraft auxiliary power units (APU) commonly provide pressurized air and controlled speed shaft power to the aircraft systems as an alternative to extracting this energy from the main engine compressor flow and accessory gearboxes. The APU is often used to power systems when the main engines are shut down.

Known ground-based APUs typically include added weight and complexity which may not be compatible with use in aircraft applications.

SUMMARY

In one aspect, there is provided an auxiliary power unit (APU) for an aircraft, the APU comprising: at least one internal combustion engine in driving engagement with an engine shaft; a generator having a generator shaft in driving engagement with the engine shaft to provide electrical power for the aircraft, a compressor having an outlet in fluid communication with an inlet of the at least one internal combustion engine; and a turbine having an inlet in fluid communication with an outlet of the at least one internal combustion engine.

In another aspect, there is provided an engine assembly for use as an auxiliary power unit for an aircraft, the engine assembly comprising: at least one internal combustion engine in driving engagement with an engine shaft; a generator having a generator shaft in driving engagement with the engine shaft to provide electrical power for the aircraft; a compressor having an outlet in fluid communication with an inlet of the at least one internal combustion engine; a first stage turbine having an inlet in fluid communication with an outlet of the at least one internal combustion engine; and a second stage turbine having an inlet in fluid communication with an outlet of the first stage turbine; wherein at least one of the turbines is in driving engagement with the compressor.

In a further aspect, there is provided a method of providing electrical power to an aircraft, the method comprising: flowing compressed air from an outlet of a compressor to an inlet of at least one internal combustion engine of an engine assembly; rotating an engine shaft with the at least one internal combustion engine; driving a generator providing electrical power to the aircraft with the engine shaft through a drive engagement between a shaft of the generator and the engine shaft; and driving a turbine of the engine assembly with exhaust from the at least one internal combustion engine.

DESCRIPTION OF THE DRAWINGS

Reference is now made to the accompanying figures in which:

FIG. **1** is a schematic view of a compound engine assembly in accordance with a particular embodiment;

FIG. **2** is a cross-sectional view of a Wankel engine which can be used in a compound engine assembly such as shown in FIG. **1**, in accordance with a particular embodiment;

FIGS. **3-5** are schematic views of flow distribution assemblies which can be used in a compound engine assembly such as shown in FIG. **1**, in accordance with particular embodiments;

FIGS. **6-7** are schematic views of cooling assemblies which can be used in a compound engine assembly such as shown in FIG. **1**, in accordance with particular embodiments;

FIG. **8** is a schematic view of a compound engine assembly in accordance with a particular embodiment;

FIG. **9** is a schematic view of a compound engine assembly in accordance with another particular embodiment, which may be used with the flow distribution assemblies of FIGS. **3-5** and/or the cooling assemblies of FIGS. **6-7**;

FIG. **10** is a schematic view of an engine assembly in accordance with another particular embodiment, which may be used with the flow distribution assemblies of FIGS. **3-5** and/or the cooling assemblies of FIGS. **6-7**; and

FIG. **11** is a schematic view of a gas turbine engine in accordance with a particular embodiment.

DETAILED DESCRIPTION

Referring to FIG. **1**, a compound engine assembly **10** is schematically shown. The compound engine assembly **10** is particularly, although no exclusively, suitable for use as an airborne auxiliary power unit (APU). The compound engine assembly **10** includes an engine core **12** having an engine shaft **16** driving a load, shown here as a generator, for example to provide electrical power to an aircraft. Other possible loads may include, but are not limited to, a drive shaft, accessories, rotor mast(s), a compressor, or any other type of load or combination thereof. The compound engine assembly **10** further includes a compressor **18**, a turbine section **20** compounding power with the engine core **12** and in driving engagement with the compressor **18** and generally including a first stage turbine **22** and a second stage turbine **24**, and a flow distribution assembly **25**, **125**, **225**, examples of which will be described further below.

In a particular embodiment, the engine core **12** includes one or more rotary engine(s) drivingly engaged to the common shaft **16** driving the load and each having a rotor sealingly engaged in a respective housing, with each rotary type engine having a near constant volume combustion phase for high cycle efficiency. The rotary engine(s) may be Wankel engine(s). Referring to FIG. **2**, an exemplary embodiment of a Wankel engine is shown. Each Wankel engine comprises a housing **32** defining an internal cavity with a profile defining two lobes, which is preferably an epitrochoid. A rotor **34** is received within the internal cavity. The rotor defines three circumferentially-spaced apex portions **36**, and a generally triangular profile with outwardly arched sides. The apex portions **36** are in sealing engagement with the inner surface of a peripheral wall **38** of the housing **32** to form three working chambers **40** between the rotor **34** and the housing **32**.

The rotor **34** is engaged to an eccentric portion **42** of the shaft **16** to perform orbital revolutions within the internal cavity. The shaft **16** performs three rotations for each orbital revolution of the rotor **34**. The geometrical axis **44** of the rotor **34** is offset from and parallel to the axis **46** of the housing **32**. During each orbital revolution, each chamber **40** varies in volume and moves around the internal cavity to undergo the four phases of intake, compression, expansion and exhaust.

An intake port **48** is provided through the peripheral wall **38** for successively admitting compressed air into each working chamber **40**. An exhaust port **50** is also provided through the peripheral wall **38** for successively discharging the exhaust gases from each working chamber **40**. Passages **52** for a glow plug, spark plug or other ignition element, as well as for one or more fuel injectors (not shown) are also provided through the peripheral wall **38**. Alternately, the intake port **48**, the exhaust port **50** and/or the passages **52** may be provided through an end or side wall **54** of the housing; and/or, the ignition element and a pilot fuel injector may communicate with a pilot subchamber (not shown) defined in the housing **32** and communicating with the internal cavity for providing a pilot injection. The pilot subchamber may be for example defined in an insert (not shown) received in the peripheral wall **38**.

In a particular embodiment the fuel injectors are common rail fuel injectors, and communicate with a source of Heavy fuel (e.g. diesel, kerosene (jet fuel), equivalent biofuel), and deliver the heavy fuel into the engine(s) such that the combustion chamber is stratified with a rich fuel-air mixture near the ignition source and a leaner mixture elsewhere.

For efficient operation the working chambers **40** are sealed, for example by spring-loaded apex seals **56** extending from the rotor **34** to engage the peripheral wall **38**, and spring-loaded face or gas seals **58** and end or corner seals **60** extending from the rotor **34** to engage the end walls **54**. The rotor **34** also includes at least one spring-loaded oil seal ring **62** biased against the end wall **54** around the bearing for the rotor **34** on the shaft eccentric portion **42**.

Each Wankel engine provides an exhaust flow in the form of a relatively long exhaust pulse; for example, in a particular embodiment, each Wankel engine has one explosion per 360° of rotation of the shaft, with the exhaust port remaining open for about 270° of that rotation, thus providing for a pulse duty cycle of about 75%. By contrast, a piston of a reciprocating 4-stroke piston engine typically has one explosion per 720° of rotation of the shaft with the exhaust port remaining open for about 180° of that rotation, thus providing a pulse duty cycle of 25%.

In a particular embodiment which may be particularly but not exclusively suitable for low altitude, each Wankel engine has a volumetric expansion ratio of from 5 to 9, and a volumetric compression ratio lower than the volumetric expansion ratio. The power recovery of the first stage turbine may be maximized by having the exhaust gas temperatures at the material limit, and as such is suitable for such relatively low volumetric compression ratios, which may help increase the power density of the Wankel engine and may also improve combustion at high speed and of heavy fuel.

It is understood that other configurations are possible for the engine core **12**. The configuration of the engine(s) of the engine core **12**, e.g. placement of ports, number and placement of seals, etc., may vary from that of the embodiment shown. In addition, it is understood that each engine of the engine core **12** may be any other type of internal combustion engine including, but not limited to, any other type of rotary engine, and any other type of non-rotary internal combustion engine such as a reciprocating engine.

Referring back to FIG. **1**, the compressor **18** is a supercharger compressor which may be a single-stage device or a multiple-stage device and may be a centrifugal or axial device with one or more rotors having radial, axial or mixed flow blades. Air enters the compressor and is compressed and delivered to an outlet conduit **70** communicating with the outlet **18***o* of the compressor **18**, and then circulated in part to an inlet conduit **71** communicating with the outlet conduit **70** through the flow distribution assembly **25**, **125**, **225**. The inlet conduit **71** delivers the compressed air to the inlet **12***i* of the engine core **12**, which corresponds to or communicates with the inlet of each engine of the engine core **12**. In a particular embodiment, the flow and pressure ratio of the compressor **18** is regulated using variable inlet guide vanes (VIGV) and/or a variable diffuser at the inlet of the compressor **18** and both generally indicated at **72**, to achieve flow and power modulation. In a particular embodiment, the compressor **18** has a compression pressure ratio of approximately 4:1. Other values are also possible.

In the embodiment shown, the compressor outlet **18***o* is also in fluid communication with a bleed conduit **74** through the flow distribution assembly **25**, **125**, **225**, which provides a fluid communication between the outlet conduit **70** and the bleed conduit **74**. The bleed conduit **74** has an end configured for connection to a pneumatic system of the aircraft such that part of the compressed air from the compressor **18** may also be supplied to the aircraft to support the aircraft pneumatic system. Accordingly, the compressor **18** provides both bleed air to the aircraft and compressed air to the engine core **12**.

The engine core **12** receives the pressurized air from the compressor **18** and burns fuel at high pressure to provide energy. Mechanical power produced by the engine core **12** drives the electrical generator **14** which provides power for the aircraft; in the embodiment shown the connection between the shaft **16** of the engine core **12** and the generator **14** is done through an appropriate type of gearbox **30**. In another embodiment, the electrical generator **14** has a design speed compatible with the rotational speed of the engine core **12**, for example from about 6000 to about 10000 rpm (rotations per minute) with an engine core **12** including rotary engine(s), and the shaft **16** of the engine core **12** drives the electrical generator **14** directly (see FIG. **9**)—i.e. through any type of engagement between the engine shaft **16** with the shaft of the generator rotor resulting in both shafts rotating at a same speed. In a particular embodiment, direct driving of the electrical generator **14** may provide for a reduction in gear losses which can be around 1% of the applied load; in a particular embodiment, the applied load of the generator **14** is about 200 hp and accordingly a loss reduction of approximately 2 hp in the waste heat produced by the engine assembly **10** may be obtained.

In a particular embodiment, the engine core **12** includes rotary engine(s), for example Wankel engine(s), and the generator **14** directly driven by the engine core has a nominal frequency of 400 Hz (e.g. actual frequency range of approximately 380-420 Hz) and is a 6 pole, 3 phases, alternative current generator having a design speed of from 7600 to 8400 rpm. In another particular embodiment, the generator **14** directly driven by the rotary (e.g. Wankel) engine core has a nominal frequency of 400 Hz and is a 8 pole, 3 phases, alternative current generator having a design speed of from 5700 to 6300 rpm. In another particular embodiment, the generator **14** directly driven by the rotary (e.g. Wankel) engine core has a nominal frequency of 400 Hz and is a 4 pole, 3 phases, alternative current generator having a design speed of from 11400 to 12600 rpm.

Is it understood that other types of generators **14** may be used. For example, the engine assembly **10** used as an APU can be configured to provide other high frequency alternative current supplies by selecting the operating speed and generator pole count to provide minimum weight, volume and/or heat as required. Variable speed operation may be employed when the associated electrical load is not frequency sensitive. Other variations are also possible.

The shaft **16** of the engine core **12** is also mechanically coupled to the rotor(s) of the compressor **18** such as to provide mechanical power thereto, through another gearbox **31**. In a particular embodiment, the gearbox **31** providing the mechanical coupling between the rotor(s) of the compressor **18** and the engine core **12** defines a speed ratio of about 10:1 between the compressor rotor(s) and engine core.

In a particular embodiment where the engine core **12** includes internal combustion engine(s), each engine of the engine core **12** provides an exhaust flow in the form of exhaust pulses of high pressure hot gas exiting at high peak velocity. The outlet **12***o* of the engine core **12** (i.e. the outlet of each engine of the engine core **12**) is in fluid communication with the inlet **22***i* of the first stage turbine **22**, and accordingly the exhaust flow from the engine core **12** is supplied to the first stage turbine **22**. Mechanical energy recovered by the first stage turbine **22** is coupled to the shaft **16** of the engine core **12** via a gearbox **33**; the rotor(s) of the compressor **18** are thus drivingly engaged to the rotor(s) of the first stage turbine **22** through the engine core **12**. In a particular embodiment, the first stage turbine **22** is configured as a velocity turbine, also known as an impulse turbine, and recovers the kinetic energy of the core exhaust gas while creating minimal or no back pressure. The first stage turbine **22** may be a centrifugal or axial device with one or more rotors having radial, axial or mixed flow blades.

The inlet **24***i* of the second stage turbine **24** is in fluid communication with the outlet **22***o* of the first stage turbine **22** and completes the recovery of available mechanical energy from the exhaust gas. The second turbine **24** is also coupled to the shaft **16** of the engine core **12** through the gearbox **33**; the rotor(s) of the compressor **18** are thus drivingly engaged to the rotor(s) of the second stage turbine **24** through the engine core **12**. In a particular embodiment, the second stage turbine **24** is configured as a pressure turbine, also known as a reaction turbine. The second stage turbine **24** may be a centrifugal or axial device with one or more rotors having radial, axial or mixed flow blades.

In the embodiment shown, the rotors of the first and second stage turbines **22**, **24** are connected to a same shaft **23** which is coupled to the engine core **12** through the gearbox **33**. Alternately, the turbines **22**, **24** could be mounted on different shafts, for example with the first stage turbine **22** mounted on a first shaft coupled to the engine shaft **16** (for example through the gearbox **23**) and the second stage turbine **24** mounted on a second shaft drivingly engaged to the compressor **18**.

A pure impulse turbine works by changing the direction of the flow without accelerating the flow inside the rotor; the fluid is deflected without a significant pressure drop across the rotor blades. The blades of the pure impulse turbine are designed such that in a transverse plane perpendicular to the direction of flow, the area defined between the blades is the same at the leading edges of the blades and at the trailing edges of the blade: the flow area of the turbine is constant, and the blades are usually symmetrical about the plane of the rotating disc. The work of the pure impulse turbine is due only to the change of direction in the flow through the turbine blades. Typical pure impulse turbines include steam and hydraulic turbines.

In contrast, a reaction turbine accelerates the flow inside the rotor but needs a static pressure drop across the rotor to enable this flow acceleration. The blades of the reaction turbine are designed such that in a transverse plane perpendicular to the direction of flow, the area defined between the blades is larger at the leading edges of the blades than at the trailing edges of the blade: the flow area of the turbine reduces along the direction of flow, and the blades are usually not symmetrical about the plane of the rotating disc. The work of the pure reaction turbine is due mostly to the acceleration of the flow through the turbine blades.

Most aeronautical turbines are not "pure impulse" or "pure reaction", but rather operate following a mix of these two opposite but complementary principles—i.e. there is a pressure drop across the blades, there is some reduction of flow area of the turbine blades along the direction of flow, and the speed of rotation of the turbine is due to both the acceleration and the change of direction of the flow. The degree of reaction of a turbine can be determined using the temperature-based reaction ratio (equation 1) or the pressure-based reaction ratio (equation 2), which are typically close to one another in value for a same turbine:

$$\text{Reaction}(T) = \frac{(t_{S3} - t_{S5})}{(t_{S0} - t_{S5})} \quad (1)$$

$$\text{Reaction}(P) = \frac{(P_{S3} - P_{S5})}{(P_{S0} - P_{S5})} \quad (2)$$

where T is temperature and P is pressure, s refers to a static port, and the numbers refers to the location the temperature or pressure is measured: 0 for the inlet of the turbine vane (stator), 3 for the inlet of the turbine blade (rotor) and 5 for the exit of the turbine blade (rotor); and where a pure impulse turbine would have a ratio of 0 (0%) and a pure reaction turbine would have a ratio of 1 (100%).

In a particular embodiment, the first stage turbine **22** is configured to take benefit of the kinetic energy of the pulsating flow exiting the engine core **12** while stabilizing the flow, and the second stage turbine **24** is configured to extract energy from the remaining pressure in the flow while expanding the flow. Accordingly, the first stage turbine **22** has a smaller reaction ratio than that of the second stage turbine **24**.

In a particular embodiment, the second stage turbine **24** has a reaction ratio higher than 0.25; in another particular embodiment, the second stage turbine **24** has a reaction ratio higher than 0.3; in another particular embodiment, the second stage turbine **24** has a reaction ratio of about 0.5; in another particular embodiment, the second stage turbine **24** has a reaction ratio higher than 0.5.

In a particular embodiment, the first stage turbine **22** has a reaction ratio of at most 0.2; in another particular embodiment, the first stage turbine **22** has a reaction ratio of at most 0.15; in another particular embodiment, the first stage turbine **22** has a reaction ratio of at most 0.1; in another particular embodiment, the first stage turbine **22** has a reaction ratio of at most 0.05.

It is understood that any of the above-mentioned reaction ratios for the second stage turbine **24** can be combined with any of the above-mentioned reaction ratios for the first stage turbine **22**, and that these values can correspond to pressure-based or temperature-based ratios. Other values are also possible. For example, in a particular embodiment, the two turbines **22**, **24** may have a same or similar reaction ratio; in another embodiment, the first stage turbine **22** has a higher reaction ratio than that of the second stage turbine **24**. Both turbines **22**, **24** may be configured as impulse turbines, or both turbines **22**, **24** may be configured as pressure turbines.

Is it understood that the connections between the rotors of the compressor **18** and turbines **22**, **24** may be different than the embodiment shown. For example, the rotors of the compressor **18** and turbines **22**, **24** may be coupled to the engine core **12** by a gearing system or variable speed drive such that power can be shared mechanically. Alternately, the compressor may be a turbocharger directly driven by the second stage turbine **24** without power transfer between the compressor **18** and the engine core **12**, for example by having the rotors of the compressor **18** and second stage turbine **24** mounted on common shaft rotating independently of the shaft **16** of the engine core **12**. In this case a variable area turbine vane may be provided at the inlet of the second stage (turbocharger) turbine **24** to provide adequate control of the compressor drive.

In use, there are typically operational situations where the aircraft cannot accept compressed air from the APU but still requires the APU to run, for example to power the generator **14**. In this case the compressor **18** produces excess flow and needs to be protected from surge. In the embodiment shown, the compressor outlet **18***o* is also in fluid communication with an excess air duct **82** receiving this excess or surge flow, through the flow distribution assembly **25**, **125**, **225** which provides a fluid communication between the outlet conduit **70** and the excess air duct **82**. The excess air duct **82** provides an alternate path for the excess air produced by the compressor **18**.

In a particular embodiment which is not shown, the excess air is dumped to atmosphere, for example by having the excess air duct **82** in fluid communication with the exhaust of the engine assembly **10**. In the embodiment shown, the excess air duct **82** has a first end communicating with the compressor outlet **18***o* and an opposed end communicating with the second stage turbine inlet **24***i*, such as to recover energy from the main flow and the surge excess flow. The excess air duct **82** thus defines a flow path between the compressor outlet **18***o* and the turbine section which is separate from the engine core **12**. The excess air duct **82** may communicate with the second stage turbine inlet **24***i* together with the exhaust from the first stage turbine outlet **22***o* through an inlet mixing device, or through a partial segregated admission turbine configuration (segregated admission nozzle) where some vane passages in the turbine entry nozzle are dedicated to the flow from the excess air duct **82** while other vane passages are dedicated to the exhaust flow from the first stage turbine outlet **22***o*. The second stage turbine **24** may feature a variable nozzle to facilitate control of load sharing and different levels of returned excess air.

The excess air duct **82** may alternately communicate with the inlet **22***i* of the first stage turbine **22**, or with the inlet of a third turbine (not shown) dedicated to recovering excess air energy. Such a third turbine may be connected to the shaft **16** of the engine core **12**, for example through an overrunning clutch, to return the energy extracted from the excess flow to the shaft **16**, or may be used to drive other elements, including, but not limited to, a cooling fan and/or an additional generator. Other types of connections and configurations are also possible.

In a particular embodiment, an exhaust heat exchanger **28** is provided to provide heat exchange relationship between the air circulating through the excess air duct **82** and the exhaust air from the outlet **24***o* of the second stage turbine **24**. The heat exchanger **28** thus includes at least one first conduit **28***a* in heat exchange relationship with at least one second conduit **28***b*. The excess air duct **82** is in fluid communication with the second stage turbine inlet **24***i* through the first conduit(s) **28***a* of the heat exchanger **28**, and the second conduit(s) **28***b* of the heat exchanger **28** is/are in fluid communication with the second stage turbine outlet **24***o* such that the exhaust from the second stage turbine **24** circulates therethrough. In a particular embodiment, the exhaust heat exchanger **28** recovers energy from the waste heat in the exhaust and increases the temperature of the excess flow (surge bleed flow) coming into the second stage turbine **24**, which improves its capacity to do work in the turbine. This provides a hybrid partially recuperated cycle.

In an alternate embodiment, the exhaust heat exchanger **28** is omitted.

Exemplary embodiments for the flow diverting assembly **25**, **125**, **225** will now be described. It is understood however than the compressor outlet **18***o*/outlet conduit **70** can be in fluid communication with the engine core inlet **12***i*/inlet conduit **71**, the bleed conduit **74** and/or the excess air duct **82** through any other appropriate type or configuration of fluid communication. For example, the bleed conduit **74** could be connected to the compressor outlet **18***o* separately from the outlet conduit **70**.

Referring to FIG. **3**, in a particular embodiment, the flow diverting assembly **25** includes an intercooler **26**, and the outlet conduit **70** is connected to a branch **73** splitting the flow between a bypass conduit **78**, an intercooler inlet conduit **64** and the excess air duct **82**. The communication between the branch **73** and the excess air duct **82** is performed through a diverter valve **84** to effect throttling of the excess air flow or shut it off when required. The excess air duct **82** thus communicates with the outlet conduit **70** upstream of the intercooler **26**; in a particular embodiment, such a configuration allows for leaving maximum energy in the compressed air being diverted into the excess air duct **82**.

The intercooler **26** includes at least one first conduit **26***a* in heat exchange relationship with at least one second conduit **26***b*. Each first conduit **26***a* of the intercooler **26** has an inlet in fluid communication with the intercooler inlet conduit **64**, and an outlet in fluid communication with an intercooler outlet conduit **66**. Each second conduit **26***b* of the intercooler **26** is configured for circulation of a coolant therethrough, for example cooling air. The compressed air circulating through the first conduit(s) **26***a* is thus cooled by the coolant circulating through the second conduit(s) **26***b*.

The intercooler outlet conduit **66** is in fluid communication with the inlet conduit **71** (and accordingly with the engine core inlet **12***i*) and with the bleed conduit **74**; the communication with the bleed conduit **74** is performed through a bleed air valve **76**, which in a particular embodiment is a load control valve, to effect throttling of the bleed or shut it off when required. The intercooler **26** accordingly reduces the temperature of the compressed air going to the engine core **12** as well as the compressed air being channelled to the aircraft through the bleed air valve **76** and bleed conduit **74**. In a particular embodiment, the pre-cooling of the air going to the aircraft allows for a higher delivery pressure than APU systems which are not pre-cooled and accordingly temperature limited for safety reasons. A higher pressure delivery may generally permit smaller ducts and pneumatic equipment, which may allow for weight savings on the aircraft.

The bypass conduit **78** provides fluid communication between the outlet conduit **70** and each of the inlet conduit

71 and bleed air valve 76 in parallel of the intercooler 26, thus allowing for a selected part of the flow to bypass the intercooler 26 before reaching the bleed air valve 76 (and accordingly the bleed conduit 74) and the inlet conduit 71 (and accordingly the engine core inlet 12i). The bypass conduit 78 includes a bypass valve 80 regulating the flow bypassing the intercooler 26. Accordingly, the temperature of the compressed air circulated to the inlet and bleed conduits 71, 74 may be regulated by changing the proportion of the flow going through the intercooler 26 by controlling the proportion of the flow going through the bypass conduit 78 with the bypass valve 80. In this particular embodiment, the compressed flow circulated to the inlet conduit 71 has the same temperature as the compressed flow circulated to the bleed conduit 74. In a particular embodiment the compressed air is cooled by the intercooler 26 such that the air circulated to the bleed conduit 74 and the inlet conduit 71 has a temperature of 250° F. or lower; other values are also possible.

Referring to FIG. 4, another particular embodiment of the flow diverting assembly 125 is shown. The outlet conduit 70 is connected to the bypass conduit 78, the intercooler inlet conduit 64, the excess air duct 82 (through the diverter valve 84) and the bleed conduit 74 (through the bleed air valve 76). In this embodiment, since the bleed conduit 74 communicates with the outlet conduit 70 upstream of the intercooler 26, the compressed air is not cooled before being circulated to the bleed conduit 74.

The intercooler outlet conduit 66 is in fluid communication with the inlet conduit 71 (and accordingly with the engine core inlet 12i); the intercooler 26 thus reduces the temperature of the compressed air going to the engine core 12. The bypass conduit 78 provides fluid communication between the outlet conduit 70 and the inlet conduit 71 in parallel of the intercooler 26, thus allowing for a selected part of the flow to bypass the intercooler 26 before reaching the inlet conduit 71 (and accordingly the engine core inlet 12i). The temperature of the compressed air circulated to the inlet conduit 71 may be regulated by changing the proportion of the flow going through the intercooler 26 by controlling the proportion of the flow going through the bypass conduit 78 with the bypass valve 80 included therein. In a particular embodiment the compressed air circulating in the outlet conduit 70 and to the bleed conduit 74 has a temperature of 450° F. or lower, and the intercooler 26 cools part of the compressed air so that the air circulated to the inlet conduit 71 has a temperature of 250° F. or lower; other values are also possible. In a particular embodiment, using the intercooler 26 to cool only the portion of the compressed air circulated to the inlet conduit 71 may allow for the intercooler 26 to be significantly smaller than an intercooler also used to cool the portion of the air circulated to the bleed conduit 74, for example such as shown in FIG. 3.

Referring to FIG. 5, another particular embodiment of the flow diverting assembly 225 is shown. The outlet conduit 70 is connected to the bypass conduit 78, the intercooler inlet conduit 64, and the excess air duct 82 (through the diverter valve 84). A first intercooler 126 used as a pre-cooler has first conduit(s) 126a each having an inlet in fluid communication with the intercooler inlet conduit 64, and an outlet in fluid communication with an intercooler intermediate conduit 68.

The intercooler intermediate conduit 68 is in fluid communication with the bleed conduit 74 through the bleed air valve 76, and the bypass conduit 78 provides fluid communication between the outlet conduit 70 and a portion of the bleed conduit 74 upstream of the bleed air valve 76 in parallel of the intercooler 126. Accordingly, the temperature of the compressed air circulated to the bleed conduit 74 may be regulated by changing the proportion of the flow going through the intercooler 126 by controlling the proportion of the flow going through the bypass conduit 78 with the bypass valve 80.

The flow diverting assembly 225 includes a second intercooler 226 also having first conduit(s) 226a in heat exchange relationship with second conduit(s) 226b. Each first conduit 226a of the intercooler 226 has an inlet in fluid communication with the intercooler intermediate conduit 68, and an outlet in fluid communication with the intercooler outlet conduit 66. Each second conduit 226b of the intercooler 226 is configured for circulation of a coolant therethrough, for example cooling air.

The intercooler outlet conduit 66 is in fluid communication with the inlet conduit 71 (and accordingly with the engine core inlet 12i); the second intercooler 226 thus further reduces the temperature of the compressed air going to the engine core 12. An additional bypass conduit 178 provides fluid communication between the intercooler intermediate conduit 68 and the inlet conduit 71 in parallel of the intercooler 226, thus allowing for a selected part of the flow to bypass the intercooler 226 before reaching the inlet conduit 71 (and accordingly the engine core inlet 12i). The additional bypass conduit 178 includes an additional bypass valve 180 to regulate the flow circulating therethrough. The temperature of the compressed air circulated to the inlet conduit 71 may be regulated by changing the proportion of the flow going through the intercoolers 126, 226 by controlling the proportion of the flow going through the bypass conduits 78, 178 with the bypass valves 80, 180.

An alternate embodiment is shown in dotted lines, where the bypass conduit 178 is replaced by a bypass conduit 178' containing bypass valve 180' and extending between the outlet conduit 70 and inlet conduit 71.

The inlet conduit 71 thus communicates with the pre-cooler intercooler 126 at least in part through the second intercooler 226, while the bleed conduit 74 communicates with the pre-cooler intercooler 126 upstream of the second intercooler 226, thus independently thereof. The arrangement thus allows for separate regulation of the temperature of the flow reaching the bleed conduit 74 and of the flow reaching the inlet conduit 71. For example, the proportion of the flow circulating through the intercooler 126 may be selected such that the temperature of the flow reaching the bleed conduit 74 is 450° F. or lower, and the temperature of the flow is further reduced in the second intercooler 226 to have a value of 250° F. or lower when reaching the inlet conduit 71. Other values are also possible.

In all embodiments, the flow diverting assembly 25, 125, 225 may include pressure, temperature and/or flow sensors, and/or closed loop system(s) controlling the position of one, some or all of the valves 76, 80, 84, 180, 180'. Any one, some or all of the valves 76, 80, 84, 180, 180' may be a hydraulically, pneumatically or electrically driven modulating valve.

In a particular embodiment, the engine assembly 10 is air startable from the pneumatic system of the aircraft or the engine bleed, as opposed to electrical power. Opening the bleed air valve 76 and the diverter valve 84 admits pressurized air to the second stage turbine 24, thereby providing a means to start rotation of the engine assembly 10. Such a configuration may thus allow for a rapid start in flight without the need to use electrical power. When provided, the third turbine (not shown) receiving air from the excess air duct 82 could also allow for air starting of the engine assembly **10**. Check valves or bypass valves (not shown) may be needed to prevent reverse flow through other parts of the engine assembly **10**. In both cases the compressor **18** is dead headed on the bleed side, so the engine start at low speed, and the external start flow is cancelled as soon as possible before accelerating to full speed to prevent high energy compressor stalls.

In an alternate embodiment, the diverter valve **84** is omitted or may be simplified to a two position on/off valve. The flow from the compressor **18** is ducted to the engine core **12** and to the excess air duct **82**, and the compressed air is "bled" from the excess air duct **82** as required by the aircraft, limited if necessary by the load control valve **76**. Such a configuration may allow loss reduction by eliminating the regulating or diverter valve pressure drop between the compressor **18** and the downstream heat exchanger **28** and turbine. When a two position diverter valve **84** is employed the valve is closed when the aircraft has a high pneumatic demand on the engine assembly **10** and fully open when the engine assembly **10** is operated with low pneumatic demand or for electrical power only.

In an alternate embodiment, the intercoolers **26**, **126**, **226** are omitted and the flow is circulated to the inlet conduit **71** and bleed conduit **74** without being cooled.

Referring to FIG. **6**, in a particular embodiment, the engine assembly **10** includes an oil cooler **88** to remove heat from the oil system of the engine assembly **10**, and an engine core liquid cooler **89** to remove heat from the coolant (e.g. water, oil or other liquid coolant) of the cooling system of the engine core **12**. A coolant pump **94** circulates the coolant between the engine core **12** and the engine core liquid cooler **89**. The coolers **88**, **89** are integrated with the intercooler **26**/**226** (and pre-cooler intercooler **126** if provided) in a cooling assembly to prevent replication of items like cooling fans and eductors. In this particular embodiment, the coolers **88**, **89** and intercoolers **26**/**226**, **126** are arranged in series in a single air duct **90** which is vented by a cooling fan **92** in ground operations or part of a ram air circuit in flight. The cooling fan **92** may be driven by any suitable rotating element of the engine assembly **10**, or powered by the generator **14**. A single inlet and exhaust can thus be used for providing coolant to all the coolers **88**, **89**, and intercoolers **26**/**226**, **126**. The coolers **88**, **89** and intercooler **26**/**226**, **126** are placed within the duct according to their temperature requirements. In the embodiment shown, the oil cooler **88** and engine core liquid cooler **89** have the lowest temperature requirements (e.g. requiring to cool the fluids therein at around 180° F. to 200° F.), and the cooling air temperature at the inlet of the air duct **90** is 130° F. or lower; the intercooler **26**/**226** has a higher temperature requirement than the coolers **88**, **89** (e.g. around 250° F.), and the pre-cooler intercooler **126** (if provided) has a higher temperature requirement than the intercooler **26**/**226** (e.g. around 450° F.). Other values are also possible.

Referring to FIG. **7**, another embodiment for the cooling assembly is shown. In this embodiment, the coolers **88**, **89** and the intercooler **26**/**226** are in parallel in the air duct **190** vented by the cooling fan **92**, with the pre-cooler intercooler **126** being provided downstream of the others. In another embodiment which is not shown, the coolers **88**, **89** and intercoolers **26**/**226**, **126** are all placed in parallel in the air duct.

Referring to FIG. **8**, a compound engine assembly **110** with cooling assembly in accordance with a particular embodiment is shown. In this embodiment, the second conduit(s) **226***b* of the intercooler **226** are in fluid communication with the liquid cooling system of the engine core **20** such that the coolant from the liquid cooling system is circulated in the second conduit(s) **226***b* to cool the compressed air circulated in the first conduit(s) **226***a*. In this embodiment, a mechanical drive **96** is provided between the cooling fan **92** and the shaft **16** of the engine core **12**. The coolers **88**, **89** are placed in parallel in the cooling air duct **290** upstream of the fan **92**, and the pre-cooler intercooler **126** is placed in the duct **290** downstream of the fan **92**. In a particular embodiment, such an arrangement provides for an optimal cooling delta T to the oil, engine coolant and compressed air while preserving acceptable entry temperatures to the fan **92** to keep its power below a desirable threshold and avoid the need for more expensive materials which may be required if the fan **92** was located in a hotter zone downstream of the intercooler **126**.

Referring to FIG. **9**, a compound engine assembly **210** according to another embodiment is shown, where components similar to that of the embodiment shown in FIG. **1** are identified by the same reference numerals and are not further described herein. As described above, the engine core **12** includes one or more internal combustion engine(s) including, but not limited to, any type of rotary engine (e.g. Wankel engine), and any type of non-rotary internal combustion engine such as a reciprocating engine. Any one of the flow distribution assemblies **25**, **125**, **225** or any other appropriate flow distribution assembly may be used to distribute the flow from the outlet conduit **70** to the inlet conduit **71**, bleed conduit **74** and excess air duct **84**.

In this embodiment, the shaft **16** of the engine core **12** is only mechanically coupled to the generator **14**, and not to the rotors of the compressor **18** and turbines **22**, **24**. The first and second stage turbines **22**, **24** are mechanically coupled to the compressor **18**, for example by having their rotors supported by a same turbine shaft **123**. The first and second stage turbines **22**, **24** are also mechanically coupled to a second generator/electrical motor **114**. In a particular embodiment, the shaft **16** of the engine core **12** and the turbine shaft **123** are each coupled to their respective generator **14**, **114** through a direct connection; the generator **14** coupled to the engine core **12** may thus have a lower speed of rotation than the generator **114** coupled to the turbine shaft **123**. Alternately, one or both connection(s) may be performed through a respective gearbox (not shown).

In another particular embodiment, the generator **114** directly driven by the turbines **22**, **24** (or by one of the turbines in an embodiment where the turbines are on different shafts) has a nominal frequency of at least 400 Hz suitable for high power density aircraft electrical equipment, for example a 2 poles, alternative current generator with a nominal frequency of 400 Hz having a design speed of from 22800 to 25300 rpm, which may correspond to a nominal speed of 24000 rpm. Such a generator **114** may be used in combination with any of the particular generators **14** mentioned above.

Power from the two shafts **16**, **123** is compounded through electrical power being transferred between the two generators **14**, **114**. For example, the first generator **14** transfers power to the second generator/motor **114** which acts as a motor to drive the rotor(s) of the compressor **18**. The generators **14**, **114** also provide electrical power for the aircraft.

In the embodiment shown, a power controller **86** is provided to control power transfer between the two generators **14**, **114** and power provided to the aircraft. In a particular embodiment, the power controller **86** allows for the compressor and engine core speed ratio to be variable, with each rotational speed being scheduled independently for optimal performance. The portion of the power from the first generator **14** being transferred to the second generator/motor **114** can be controlled to achieve the most advantageous rotational speed for the rotor(s) of the compressor **18**. In addition, when the turbines **22**, **24** coupled to the compressor **18** generate excess energy, the second generator/motor **114** can also provide power to the aircraft and/or to the first generator **14** which may also act as a motor. The power controller **86** may also contain features such as frequency and voltage regulation to manage AC power quality supplied to the airframe.

Although not shown, the transfer of power from the engine core **12** to the supercharger (compressor **18** and turbines **22**, **24**) may also be performed through hydraulic or mechanical CVT systems to allow for independent speed scheduling.

Referring to FIG. **10**, an engine assembly **310** according to another embodiment is shown, where components similar to that of the embodiment shown in FIG. **1** are identified by the same reference numerals and are not further described herein.

In this embodiment, the shaft **16** of the engine core **12** is mechanically coupled to the generator **14**, and the power of the turbine(s) is not compounded with that of the engine core **12**. Although a single turbine **322** is shown, multiple turbines may be provided. The turbine **322** is mechanically coupled to the compressor **18**, for example by having their rotors supported by a same shaft **123**. The turbine **322** may be configured as an impulse turbine or as a pressure turbine, and may have any suitable reaction ratio, including, but not limited to, the ratios described above for turbines **22**, **24**. In a particular embodiment, the turbine **322** is replaced by first and second stages turbines **22**, **24** as previously described.

Any one of the flow distribution assemblies **25**, **125**, **225** or any other appropriate flow distribution assembly may be used to distribute the flow from the outlet conduit **70** to the inlet conduit **71**, bleed conduit **74** and excess air duct **84**. When more than one turbine is provided, the flow from the excess air duct **84** may be circulated to the inlet of any one of the turbines.

Although not shown, the turbine(s) **322** may also drive a separate generator or any other appropriate type of accessory. Although not shown, a power controller may be provided to control power transfer between the generator **14** and any system receiving electrical power from the generator **14**.

The above description is meant to be exemplary only, and one skilled in the art will recognize that changes may be made to the embodiments described without departing from the scope of the invention disclosed. For example, although the compressor **18** has been shown as providing compressed air both for the engine core **12** and for the aircraft, alternately the compressor **18** may be configured to act only as a supercharger for the engine core **12**, and a separate load compressor may be configured to provide the aircraft air. Such a load compressor may be driven by the engine core **12** and/or the turbines **22**, **24**, **322** either directly or through a gearbox. The two compressors may have a common inlet. Moreover, although the engine core **12** has been described as including one or more internal combustion engines, the engine core **12** may alternately be any other type of engine core in which the compressed air is mixed with fuel and ignited for generating hot combustion gases, including, but not limited to, a gas turbine engine combustor; as non-limiting examples, the intercooler **26**, **126** cooling compressed air circulated to the aircraft, and the circulation of the excess air from the compressor **18** to a turbine **22**, **24** with the excess air duct **82** to provide additional work with or without an exhaust heat exchanger **28** between the excess air and the turbine exhaust, may be applied to a gas turbine engine with a combustor, as schematically illustrated in FIG. **11**. Other modifications which fall within the scope of the present invention will be apparent to those skilled in the art, in light of a review of this disclosure, and such modifications are intended to fall within the appended claims.

The invention claimed is:

1. A power unit for an aircraft, the power unit comprising:

at least one internal combustion engine in driving engagement with an engine shaft;

a generator having a generator shaft in driving engagement with the engine shaft to provide electrical power for the aircraft, the generator and engine shafts being connected via a gearbox such as to be rotatable at a different speed;

a compressor having an outlet in fluid communication with an inlet of the at least one internal combustion engine; and

a turbine having an inlet in fluid communication with an outlet of the at least one internal combustion engine.

2. The power unit as defined in claim **1**, wherein the turbine is configured to compound power with the at least one internal combustion engine.

3. The power unit as defined in claim **1**, further comprising a bleed conduit having an end configured for connection with a pneumatic system of the aircraft, the bleed conduit in fluid communication with the outlet of the compressor through a bleed air valve selectively opening and closing the fluid communication between the outlet of the compressor and the end of the bleed conduit configured for connection to the pneumatic system.

4. The power unit as defined in claim **1**, wherein each of the at least one internal combustion engine includes a rotor sealingly and rotationally received within a respective internal cavity to provide rotating chambers of variable volume in the respective internal cavity, the rotor having three apex portions separating the rotating chambers and mounted for eccentric revolutions within the respective internal cavity, the respective internal cavity having an epitrochoid shape with two lobes.

5. The power unit as defined in claim **1**, wherein the generator has a nominal rotational speed of at least 5700 rotations per minute.

6. The power unit as defined in claim **1**, wherein the generator is selected from the group consisting of: a 6 pole, 3 phases, alternative current generator having a design speed of from 7600 to 8400 rotations per minute; a 8 pole, 3 phases, alternative current generator having a design speed of from 5700 to 6300 rotations per minute; and a 4 pole, 3 phases, alternative current generator having a design speed of from 11400 to 12600 rotations per minute.

7. The power unit as defined in claim **1**, wherein the generator is a first generator, the turbine includes at least one rotor engaged on a turbine shaft rotatable independently of the engine shaft, the assembly further comprising a second generator having a second generator shaft directly engaged to the turbine shaft such that the turbine and second generator shafts are rotatable at a same speed.

8. The power unit as defined in claim **1**, wherein the turbine is a first stage turbine, the assembly further comprising a second stage turbine having an inlet in fluid communication with an outlet of the first stage turbine.

9. The power unit as defined in claim **8**, wherein the first stage turbine is configured as an impulse turbine with a pressure-based reaction ratio having a value of at most 0.25, the second stage turbine having a higher reaction ratio than that of the first stage turbine.

10. The power unit as defined in claim **1**, further comprising variable inlet guide vanes, a variable diffuser or a combination thereof at an inlet of the compressor.

11. An engine assembly for use as a power unit for an aircraft, the engine assembly comprising:

at least one internal combustion engine in driving engagement with an engine shaft;

a generator having a generator shaft in driving engagement with the engine shaft to provide electrical power for the aircraft;

a gearbox drivingly connecting the engine shaft to the generator shaft;

a compressor having an outlet in fluid communication with an inlet of the at least one internal combustion engine;

a first stage turbine having an inlet in fluid communication with an outlet of the at least one internal combustion engine; and

a second stage turbine having an inlet in fluid communication with an outlet of the first stage turbine;

wherein at least one of the turbines is in driving engagement with the compressor.

12. The engine assembly as defined in claim **11**, further comprising a bleed conduit having an end configured for connection to a system of the aircraft, the bleed conduit being in fluid communication with the outlet of the compressor, and a bleed air valve selectively opening and closing the fluid communication between the end of the bleed conduit and the outlet of the compressor.

13. The engine assembly as defined in claim **11**, wherein each of the at least one internal combustion engine includes a rotor sealingly and rotationally received within a respective internal cavity to provide rotating chambers of variable volume in the respective internal cavity, the rotor having three apex portions separating the rotating chambers and mounted for eccentric revolutions within the respective internal cavity, the respective internal cavity having an epitrochoid shape with two lobes.

14. The engine assembly as defined in claim **11**, wherein the generator has a nominal frequency of 400 Hz.

15. The engine assembly as defined in claim **11**, wherein the generator is selected from the group consisting of: a 6 pole, 3 phases, alternative current generator having a design speed of from 7600 to 8400 rotations per minute; a 8 pole, 3 phases, alternative current generator having a design speed of from 5700 to 6300 rotations per minute; and a 4 pole, 3 phases, alternative current generator having a design speed of from 11400 to 12600 rotations per minute.

16. The engine assembly as defined in claim **11**, wherein rotors of the first and second stage turbines and of the compressor are engaged on a turbine shaft rotatable independently of the engine shaft, the generator is a first generator, the assembly further comprising a second generator having a second generator shaft directly engaged to the turbine shaft such that the turbine and second generator shafts are rotatable at a same speed.

17. The engine assembly as defined in claim **11**, wherein the first stage turbine is configured as an impulse turbine with a pressure-based reaction ratio having a value of at most 0.25, the second stage turbine having a reaction ratio higher than that of the first stage turbine.

18. A method of providing electrical power to an aircraft, the method comprising:

flowing compressed air from an outlet of a compressor to an inlet of at least one internal combustion engine of an engine assembly;

rotating an engine shaft with the at least one internal combustion engine;

driving a generator providing electrical power to the aircraft with the engine shaft through a drive engagement between a shaft of the generator and the engine shaft, the shaft of the generator being driven at a different speed than a rotational speed of the engine shaft; and

driving a turbine of the engine assembly with exhaust from the at least one internal combustion engine.

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